NASA TM-83204

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NASA Technical Memorandum 83204

NASA-TM-83204 19810024663

SIMPLIFIED OFF-DESIGN PERFORMANCE MODEL OF A DRY TURBOFAN ENGINE CYCLE

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September 1981

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ABSTRACT

The specific thrust and fuel-air ratio for a dry turbofan engine cycle were calculated for several power levels over a range of altitudes and Mach numbers. The engine has a design fan pressure ratio of 2.9, compressor pressure ratio of 8.0, and bypass ratio of 0.6. Nominal engine component curves were picked to approximate the calculated data to construct a simplified model of the off-design performance of the engine. The model was then used to construct a simplified design-point engine model for the full-power condition.

SUMMARY

The specific thrust and fuel-air ratio for a dry turbofan engine cycle were calculated for altitudes of 0 and 10 kilometers and Mach numbers of 0, 1 and 2 for a number of power settings, each specified by fan pressure ratio, compressor pressure ratio, and main burner temperature. Theoretical engine component performance curves, for which the specific heat is considered to be constant but unknown, were picked to approximate the calculated data for each engine component. The resultant curves form the basis of an off-design performance model of the engine which does not require complicated equations to calculate specific heats, entropy functions, and enthalpy functions. The parameters were then set to their full-power values and the model was reduced to a simplified design-point model for the engine.

INTRODUCTION

In the study of integrated propulsion system and flight controls, it is necessary to model the effects of varying flight conditions upon propulsion system operation. Perturbation models of engines used for multivariable control law design do not generally include these effects. The operation of an engine depends upon the air entering it (specifically, the pressure and total temperature at the face of the fan for a turbofan engine). These, in turn, are dependent on the flight Mach number, altitude, and the inlet pressure recovery ratio. Inclusion of these effects into an integrated aircraft model enables the study of interactions which may occur between flight modes such as the phugoid and variations in the engine thrust and internal operating conditions. This report describes the results of an effort which has sought to characterize the effects of flight condition and inlet pressure recovery ratio upon the operating parameters of a contemporary dry turbofan jet engine type.

N81-33206#

Two quantities which characterize the performance of a turbofan engine are the specific thrust (ST) and the fuel-air ratio (W_f/W_a). The specific thrust is the ratio of the net thrust force produced by the engine to the mass flow rate of the air entering the engine. Using this and the fuel-air ratio, the specific fuel consumption (SFC) of the engine may be computed. The specific fuel consumption is the ratio of fuel rate to the thrust force. Using this, the free-stream velocity, and the heating value of the fuel, the efficiency of the engine may be computed.

One method of calculating the performance of a turbofan engine is given in reference 1. This method involves complicated equations needed to calculate specific heats, entropy functions, and enthalpy functions. In an application where many of the parameters which define the engine are fixed or confined to narrow ranges, the use of a complicated computer program as is given in reference 1 may not be necessary. In this case, the program may be used to generate simplified functions defining the operation of the individual engine components. These functions may then be assembled into a simplified model of the whole engine.

This report uses the data calculated by the computer program of reference 1 to construct a simplified model of the engine defined by tables I and II and figure 1. The resultant model involves many nonlinear functions which have been represented by algebraic equations. The specific thrust and fuel-air ratio are functions of flight condition (Mach number and ambient temperature), inlet performance (pressure recovery ratio), fan pressure ratio, compressor pressure ratio, and burner temperature. A schematic diagram of the engine model is presented in figure 2. The nonlinear functions in the model are presented as algebraic equations in the text and in graphical form in figures 3 through 13.

The values for the fan pressure ratio, compressor pressure ratio, and burner temperature were then set to their full-power values. With these variables fixed, the engine model is reduced to a design-point performance model. The specific thrust and fuel-air ratio become functions of flight condition and inlet performance. A schematic of the design-point model is presented in figure 14. Many of the nonlinear functions in this model are the same as was discussed above. Additional nonlinear functions were derived. These are presented as algebraic equations in the text and in graphical form in figures 15 through 17.

SYMBOLS

b	bypass ratio
C_p	specific heat, J/kg-^oK
g	gravitational acceleration constant (9.81 m/sec^2)
h	altitude, m

HVF	heating value of fuel, J/kg
M	Mach number
$\overline{M}$	molecular weight, kg/kmole
P	pressure, Pa
R	gas constant, J/kg-$^{\circ}$K
R	universal gas constant (8.314 J/$^{\circ}$K-kg-mole)
$r_{1,0}$	inlet pressure recovery ratio
SFC	specific fuel consumption, kg/N-sec
ST	specific thrust, N-sec/kg
ST1	contribution of engine core to specific thrust, N-sec/kg
ST2	contribution of bypass to specific thrust, N-sec/kg
T	temperature, $^{\circ}$K
V	velocity, m/sec
V_c	velocity coefficient of the main nozzle
V_{cd}	velocity coefficient of the duct nozzle
W_c/W_a	coolant flow ratio
W_f/W_a	fuel-air ratio
δ	pressure relative to sea-level standard (101.4 kPa)
γ	ratio of specific heats
η	efficiency
θ	temperature relative to sea-level standard (288.2 $^{\circ}$K)
Δh	enthalpy change, J/kg
$\Delta\phi$	entropy change, J/kg-$^{\circ}$K

Subscripts

a	air
B	burner

C	compressor
F	fan
f	fuel
g	gas
s	static
SL	sea-level
T	turbine

Integer subscripts refer to engine stations depicted in figure 1.

0	free-stream
1	fan face
1'	fan exhaust
2	compressor exhaust
3	main burner
4	turbine exhaust
5	main nozzle entrance
6	main nozzle exit
7	duct nozzle entrance
8	duct nozzle exit

ENGINE MODEL DEVELOPMENT

The performance model of the dry turbofan engine developed in this report is depicted in figure 2. This model is based on the theoretical performance characteristics of the individual engine components of reference 1. The characteristics are obtained by manipulating the equations of reference 1 with the specific heats, C_p, held constant. The resultant equations are then fitted to data obtained by executing the program of reference 1 over a range of operating conditions. If the specific heats do not vary radically, then the derived formulae for the engine component characteristics will at least be of the proper form, if not having the precise numbers. In each case, the derived formulae are fitted to data which are calculated by the more exact program. This data is presented with the fitted formulae in figures 3 through 13.

The specific heat used in the data calculations is a fourth order polynomial function of temperature and is also a function of the fuel-air ratio of the gas being considered. Treating the specific heat to be a constant, the enthalpy change and entropy change reduce to the following approximations:

$$\Delta h = \int_{T_1}^{T_2} C_p(T)\,dT \simeq C_p(T_2 - T_1) \tag{1}$$

$$\Delta \phi = \int_{T_1}^{T_2} \frac{C_p(T)}{T}\,dT \simeq C_p \ln\left(\frac{T_2}{T_1}\right) \tag{2}$$

These approximations are used throughout the following development.

The engine data were calculated by the program of reference 1. The characteristics of the engine are given in table I. The data were calculated for Mach numbers of 0, 1 and 2 at sea level and 10 kilometers altitude. At each flight condition, the fan pressure ratio, compressor pressure ratio, and burner temperature were varied according to the schedule of table II.

The individual engine components are examined in the following sections. This development follows the derivations of reference 1. The approximations of equations 1 and 2 are applied in each instance. In addition, the effects of the fuel combustion products upon the thermodynamic properties of the gases in various parts of the engine are ignored. For each engine component, the theoretical (approximation) formula is given followed by a formula (with numbers) which has been picked to approximate the calculated data. Note that whenever the specific heat, C_p, appears in the theoretical formulae, its value is obscured by the effects of the approximations used. Those terms involving C_p are treated as unknowns in the process of fitting the theoretical formulae to the calculated data.

Temperature and Pressure Variables

The temperature and pressure variables used in reference 1 and in the theoretical formulae given below are absolute values denoted by T and P. The temperature and pressure variables used in the fitted formulae are relative to standard sea-level values and are denoted by θ and δ, respectively. The relationships between these variables are

$$\theta = \frac{T}{T_{SL}} \tag{3}$$

$$\delta = \frac{P}{P_{SL}} \tag{4}$$

where T_{SL} is the standard sea-level temperature of 288.2 ^{o}K
P_{SL} is the standard sea-level pressure of 101.4 kPa.

Engine Inlet

The variation of the standard ambient temperature of reference 2 with altitude is depicted in figure 3a. The temperature decreases linearly with altitude until it reaches 75.19 percent of its sea-level value at 11 kilometers altitude after which it is constant until 20 kilometers altitude.

The inlet velocity is a function of Mach number and ambient temperature.

$$V_0 = M_0 \sqrt{R_a \gamma T_{0s}} = 340.4 \sqrt{\theta_{0s}} \tag{5}$$

where V_0 is the airplane air speed
M_0 is the airplane Mach number
θ_{0s} is the ambient temperature
γ is the ratio of specific heats of the ambient air equal to $(1-R/C_p)^{-1}$.

The inlet total temperature ratio is a function of Mach number.

$$\frac{\theta_0}{\theta_{0s}} = 1 + \frac{R_a \gamma}{2C_p} M_0^{2} \simeq 1 + .199\ M_0^{2} \tag{6}$$

This formula is depicted in figure 3b.

The inlet total pressure ratio is a function of the total temperature ratio.

$$\frac{\delta_0}{\delta_{0s}} = \left(\frac{\theta_0}{\theta_{0s}}\right)^{\frac{C_p}{R_a}} \simeq \left(\frac{\theta_0}{\theta_{0s}}\right)^{3.518} \tag{7}$$

This formula is depicted in figure 3c.

Diffuser

The total temperature across the diffuser is constant and the total pressure ratio of the diffuser is the pressure recovery ratio, $r_{1,0}$.

$$\theta_1 = \theta_0 \tag{8}$$

$$\delta_1 = r_{1,0}\delta_0 \tag{9}$$

Fan

The enthalpy change of the fan (a measure of the fan work) is a function of the fan pressure ratio and the total temperature of the air entering the fan.

$$\frac{\Delta h_F}{\theta 1} = \frac{C_p T_{SL}}{\eta_F}\left[\left(\frac{\delta_{1'}}{\delta_1}\right)^{\frac{R_a}{C_p}} - 1\right] \simeq 353000\left[\left(\frac{\delta_{1'}}{\delta_1}\right)^{.2805} - 1\right] \tag{10}$$

where $\delta_{1'}/\delta_1$ is the fan pressure ratio

η_F is the fan efficiency.

The fan temperature ratio is a function of the enthalpy change.

$$\frac{\theta_{1'}}{\theta_1} = 1 + \frac{\Delta h_F}{C_p T_1} \simeq 1 + 3.162 \times 10^{-6}\left(\frac{\Delta h_F}{\theta_1}\right) \tag{11}$$

Equations 10 and 11 are depicted in figures 4 and 5. Since the variations in the data used for figures 4 and 5 are weak functions of the fan intake temperature, only the data for the maximum temperature (sea-level, Mach 2) and the minimum temperature (10 kilometers altitude, zero velocity) are shown on these figures.

Compressor

The enthalpy change of the compressor is a function of the compressor pressure ratio and the total temperature of the air entering the compressor.

$$\frac{\Delta h_C}{\theta_{1'}} = \frac{C_p T_{SL}}{\eta_C}\left[\left(\frac{\delta_2}{\delta_{1'}}\right)^{\frac{R_a}{C_p}} - 1\right] \simeq 3853000\left[\left(\frac{\delta_2}{\delta_{1'}}\right)^{.2654} - 1\right] \tag{12}$$

where $\delta_2/\delta_{1'}$ is the compressor pressure ratio
η_C is the compressor efficiency.

The compressor temperature ratio is a function of the enthalpy change

$$\frac{\theta_2}{\theta_{1'}} = 1 + \frac{\Delta h_C}{C_p T_{1'}} \simeq 1 + 3.062 \times 10^{-6}\left(\frac{\Delta h_C}{\theta_{1'}}\right) \tag{13}$$

Equations 12 and 13 are depicted in figures 6 and 7.

Main Burner

The fuel-air ratio is a function of the temperature change across the main burner.

$$\frac{W_f}{W_a} = \left(1 - \frac{W_c}{W_a}\right)\frac{\theta_3 - \theta_2}{\frac{HVF\eta_\beta}{C_p T_{SL}} - (\theta_3 - \theta_F)} \simeq .0006 + .0042\ (\theta_3 - \theta_2) \tag{14}$$

where W_c/W_a is the coolant flow ratio
HVF is the heating value of the fuel
η_β is the efficiency of the burner
θ_F is the temperature of the fuel.

This formula is depicted in figure 8.

Turbine

The enthalpy change of the turbine is equal to the sum of the enthalpy changes of the fan and compressor with allowance made for the higher airflow of the fan.

$$\Delta h_T = (1+b)\Delta h_F + \Delta h_C \tag{15}$$

where b is the bypass ratio.

The temperature drop across the turbine is a function of the enthalpy change.

$$\theta_3 - \theta_4 = \frac{\Delta h_t}{C_p T_{SL}\left(1 - \frac{W_c}{W_a}\right)} \simeq -.07797 + 3.220 \times 10^{-6} \Delta h_T \tag{16}$$

The pressure ratio across the turbine is a function of the temperature ratio

$$\frac{\delta_4}{\delta_3} = \left[1 + \frac{1}{\eta_T}\left(\frac{\theta_4}{\theta_3} - 1\right)\right]^{\frac{\overline{M}_g C_p}{R}} \simeq \left[1 + \frac{1}{.9}\left(\frac{\theta_4}{\theta_3} - 1\right)\right]^{4.290} \tag{17}$$

where η_T is the efficiency of the turbine.

Equations 16 and 17 are depicted in figures 9 and 10.

The exhaust gas of the turbine is a mixture of gas from the main burner and a portion of cooling air from the compressor. The temperature of the turbine exhaust is determined by equating the energy of the gas mixture to the energies of each of the components.

$$\Delta h(T_5) = \frac{W_c}{W_a}\Delta h(T_2) + \left(1 - \frac{W_c}{W_a}\right)\Delta h(T_4) \tag{18}$$

where W_c/W_a is the cooling flow ratio. A fit of the data yields

$$\theta_5 \simeq .3652 + .1010\,\theta_2 + .7856\,\theta_4 \tag{19}$$

Figure 11 illustrates the correlation between the calculated and approximated values of the turbine exhaust temperature.

Specific Thrust

The specific thrust of the engine is

$$\mathrm{ST} = \frac{(1 + W_f/W_a)V_6}{1 + b} + \frac{b\ V_8}{1 + b} - V_0 \qquad (20)$$

where V_0 is the free-stream velocity

V_6 is the exit velocity of the main nozzle

V_8 is the exit velocity of the duct nozzle

W_f/W_a is the fuel-air ratio

b is the bypass ratio.

The first term of equation 20 is the contribution to the specific thrust of the core of the engine, the second term is the contribution of the engine bypass, and the third term is the effect of the momentum of the inlet air. The specific thrust, in this paper, is defined as the amount of thrust force produced per mass flow of the total intake air of the engine and its units are newtons force per kg/sec mass rate. These three terms are computed separately as indicated in figure 2. In this figure, the first term of equation 20 is denoted by ST1 and the second term by ST2.

Main Nozzle

The component of the specific thrust produced by the main nozzle is a function of the nozzle pressure ratio and the temperature of the gas entering it.

$$\frac{\mathrm{ST1}}{\sqrt{\theta_5}} = \frac{V_c}{1 + b} \sqrt{2gC_pT_{SL}} \sqrt{1 - \left(\frac{\delta_5}{\delta_{0s}}\right)^{-\frac{R}{M_gC_p}}} \simeq 482.9 \sqrt{1 - \left(\frac{\delta_5}{\delta_{0s}}\right)^{-.2698}} \qquad (21)$$

where V_c is the main nozzle velocity coefficient. This formula is depicted in figure 12.

Duct Nozzle

The component of the specific thrust produced by the duct nozzle is a function of the nozzle pressure ratio and the temperature of the gas entering it.

$$\frac{\mathrm{ST2}}{\sqrt{\theta_{1'}}} = \frac{bV_{cd}}{1 + b} \sqrt{2gC_pT_{SL}} \sqrt{1 - \left(\frac{\delta_{1'}}{\delta_{0s}}\right)^{-\frac{R}{M_gC_p}}} \simeq 273.9 \sqrt{1 - \left(\frac{\delta_{1'}}{0s}\right)^{-2660}} \qquad (22)$$

where V_{cd} is the duct nozzle velocity coefficient. This formula is depicted in figure 13.

DESIGN-POINT MODEL

This section uses the engine performance model developed in the previous section to generate a design-point engine model. The design point (full power) condition is specified by the fan pressure ratio, the compressor pressure ratio, and the burner temperature. The fan pressure ratio sets the ratio of fan enthalpy change with fan intake temperatures $\Delta h_f/\theta_1$ (Fig. 4). This, in turn, sets the fan temperature ratio, $\theta_{1'}/\theta_1$ (Fig. 5). The compressor pressure ratio, $\delta_2/\delta_{1'}$, sets the ratio of compressor enthalpy change with compressor intake temperature, $\Delta h_c/\theta_{1'}$ (Fig. 6). This, in turn, sets the compressor temperature ratio, $\theta_2/\theta_{1'}$ (Fig. 7). These quantities and the selected burner temperature are used to reduce the engine performance model to the design-point engine model of figure 14.

The design-point model is developed for the parameters of table II corresponding to the maximum power level angle (PLA) setting.

The engine inlet and diffuser models are the same as in the previous section.

The fan model reduces to the following:

$$\frac{\Delta h_F}{\theta_1} = 122800 \tag{23}$$

$$\frac{\theta_{1'}}{\theta_1} = 1.387 \tag{24}$$

where equations 23 and 24 were obtained from the approximations of equations 10 and 11.

The compressor model reduces to the following:

$$\frac{\Delta h_c}{\theta_{1'}} = 282100 \tag{25}$$

$$\frac{\theta_2}{\theta_{1'}} = 1.864 \tag{26}$$

where equations 25 and 26 were obtained from the approximations of equations 12 and 13.

The fuel-air ratio reduces to

$$\frac{W_f}{W_a} = .02332 - .01086\ \theta_1 \tag{27}$$

where equation 27 was obtained from the approximation of equation 14 with equations 24 and 25. This formula is depicted in figure 15.

The temperature drop across the turbine becomes

$$\theta_3 - \theta_4 = -.078 + 1.903\ \theta_1 \tag{28}$$

where equation 28 was obtained by the combination of equations 15, 16, 23, 24 and 25.

The pressure ratio across the turbine becomes

$$\frac{\delta_4}{\delta_3} = [1.016 - .3908\ \theta_1]^{4.290} \tag{29}$$

where equation 29 was obtained by combining equation 28 with the approximation of equation 17. This formula is depicted in figure 16.

The turbine exhaust temperature becomes

$$\theta_5 = 4.677 - 1.234\ \theta_1 \tag{30}$$

where equation 30 was obtained by combining equations 19, 24, 26 and 28. This formula is depicted in figure 17.

The errors which are apparent in figures 15 through 17 are caused by an accumulation of small errors in the engine performance model developed in the previous section. The coefficients of equations 27, 29 and 30 were adjusted to more closely fit the calculated values. The adjusted formulae follow:

$$\frac{W_f}{W_a} = .02264 - .00988\ \theta_1 \tag{31}$$

$$\frac{\delta_4}{\delta_3} = [.9973 - .3647\ \theta_1]^{4.290} \tag{32}$$

$$\theta_5 = 4.759 - 1.272\ \theta_1 \tag{33}$$

These formulae are depicted in figures 15 through 17.

The models for the main nozzle and the duct nozzle remain the same as in the previous section.

CONCLUDING REMARKS

Two simplified performance models for a dry turbofan engine have been developed. The first model produces the specific thrust and fuel-air ratio given the flight condition, air inlet performance, fan and compressor pressure ratios, and the burner temperature. Specific values for the fan and compressor pressure ratios and the burner temperature were set in this model to produce a design-point model. Several nonlinear algebraic formulas for the performance of the individual engine components were derived by generating simplified theoretical math models which were then fitted to calculated data. Because of the relative simplicity of these models, they can be used for calculations of engine performance where computational efficiency and speed are important factors.

REFERENCES

1. Vanco, Michael R.: Computer Program for Design-Point Performance of Turbojet and Turbofan Engine Cycles. NASA TM X-1340, February 1967.

2. Anon.: U.S. Standard Atmosphere, 1962. December 1962.

Table I.- Engine Characteristics.

Variable	Description	Value	Units
η_C	Compressor Efficiency	.88	
η_T	Turbine Efficiency	.90	
η_F	Fan Efficiency	.90	
HVF	Heating Value of Fuel	46.7	mJ/kg
$r_{3,2}$	Main Burner Pressure Ratio	.95	
V_c	Main Nozzle Velocity Coefficient	.96	
V_{cd}	Duct Nozzle Velocity Coefficient	.96	
W_c/W_a	Coolant Flow Ratio	.158	
η_b	Main Burner Efficiency	.98	
PLA	Power Lever Angle	20-83	degrees
b	Bypass Ratio	0.6	
$P_{1'}/P_1$	Fan Pressure Ratio	.98-2.9	
P_2/P_1	Overall Pressure Ratio	5.54-23.0	
T_3	Main Burner Temperature	1372-1559	^oK

Table II.- Part-Power Schedules.

PLA	$P_{1'}/P_1$	$P_2/P_{1'}$	θ_3
20	.98	6.65	4.76
30	1.37	6.07	4.86
40	1.68	6.59	4.97
50	1.84	7.53	5.07
60	2.03	8.19	5.17
70	2.23	8.70	5.27
83	2.90	7.93	5.41

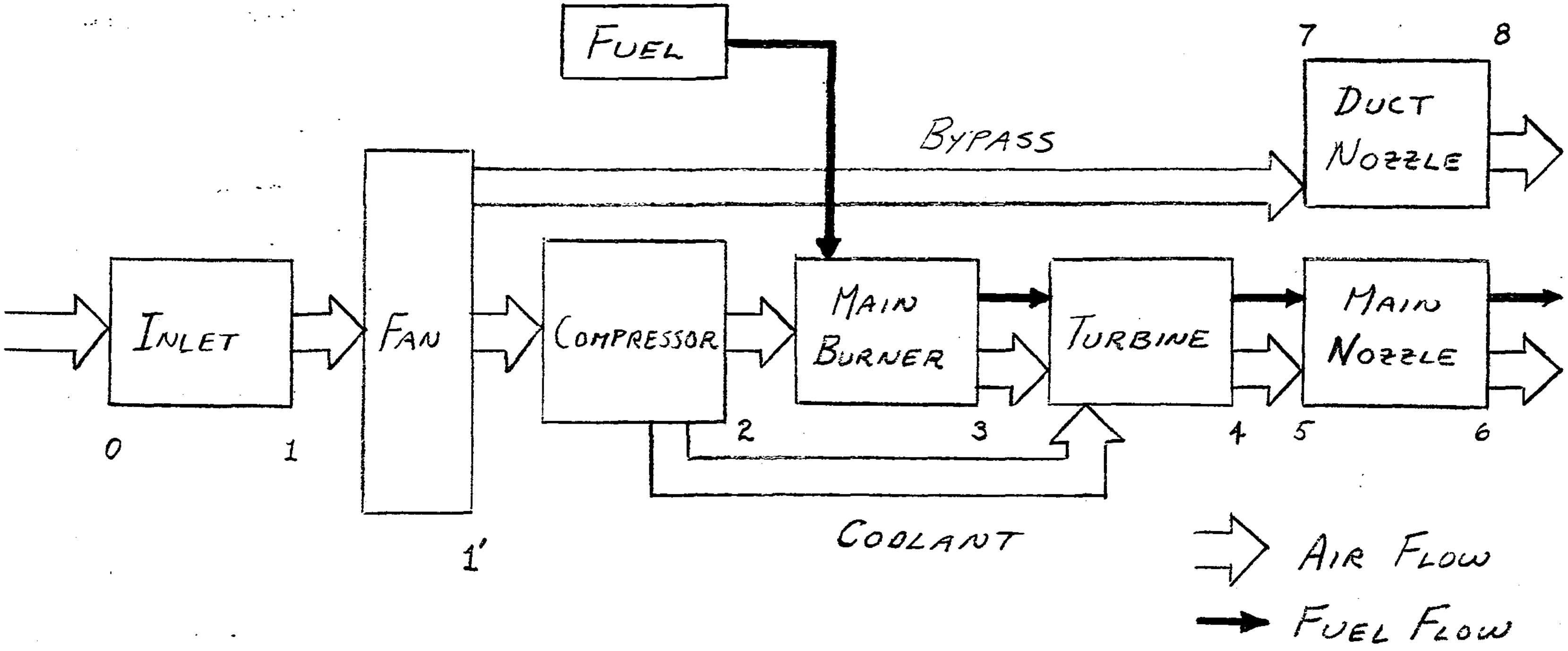

Fuel
Bypass
7
8
Duct Nozzle
Inlet
Fan
Compressor
Main Burner
Turbine
Main Nozzle
0
1
1'
2
3
4
5
6
Coolant
Air Flow
Fuel Flow

Figure 1.- Schematic Diagram of Dry Turbofan Engine.

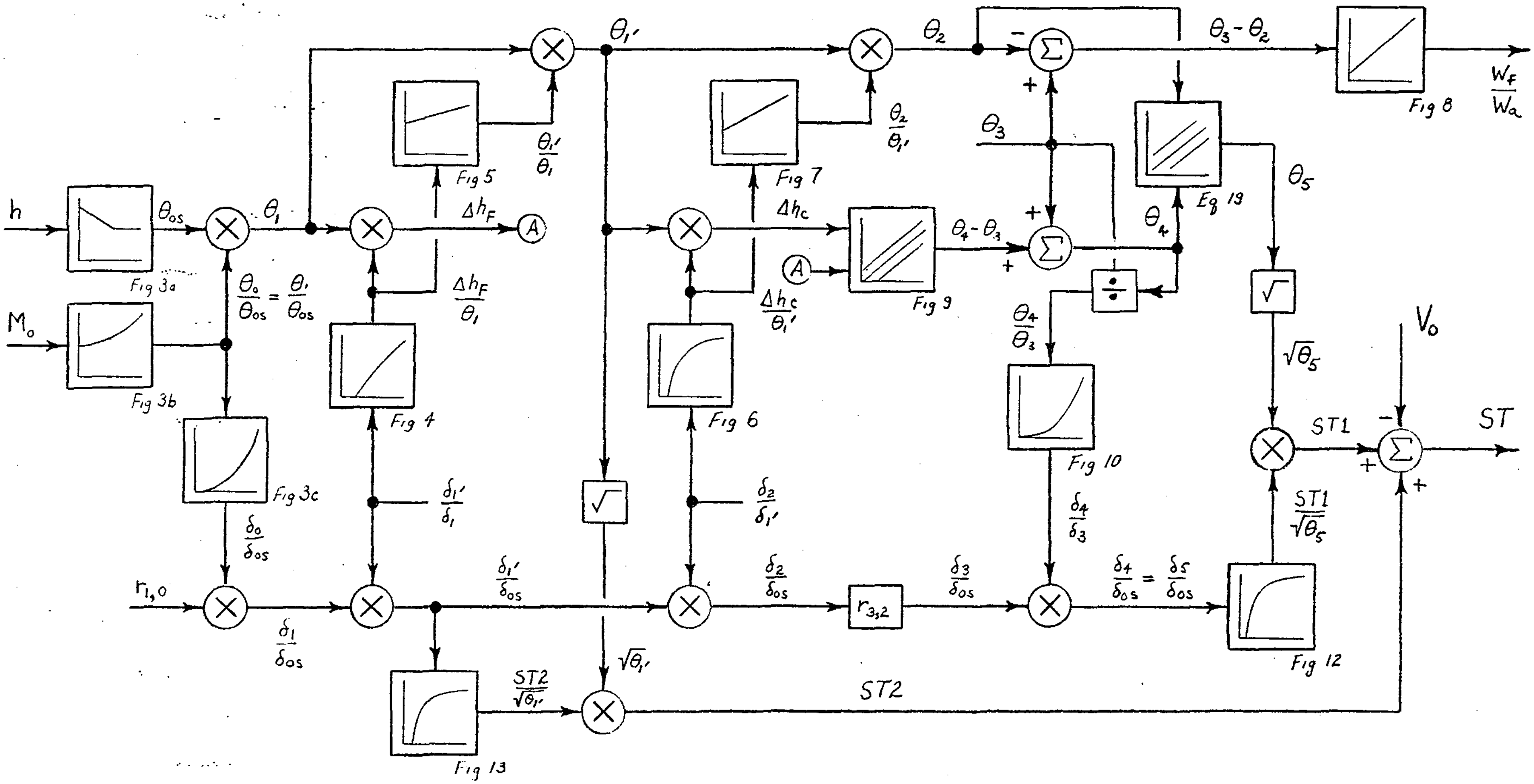
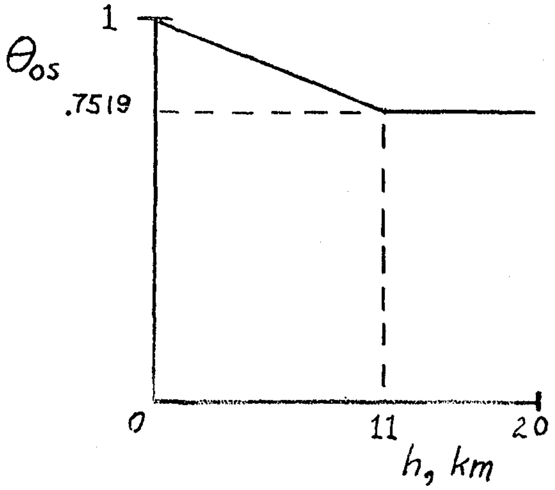
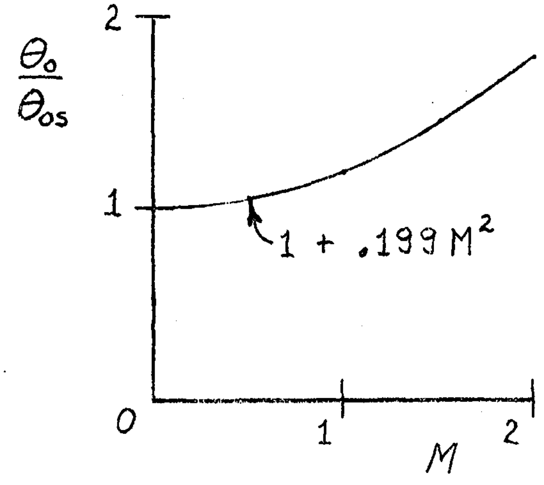

h
M₀
$r_{1,0}$
θ_{os}
θ_1
θ_1'
θ_2
$\theta_3 - \theta_2$
$\frac{W_F}{W_a}$
Fig 3a
Fig 3b
Fig 3c
$\frac{\theta_0}{\theta_{os}} = \frac{\theta_1}{\theta_{os}}$
$\frac{\delta_0}{\delta_{os}}$
$\frac{\delta_1}{\delta_{os}}$
Fig 4
Fig 5
$\frac{\theta_1'}{\theta_1}$
Δh_F
A
$\frac{\Delta h_F}{\theta_1}$
$\frac{\delta_1'}{\delta_1}$
$\frac{\delta_1'}{\delta_{os}}$
√
$\sqrt{\theta_1'}$
Fig 13
$\frac{ST2}{\sqrt{\theta_1'}}$
ST2
Fig 6
Fig 7
$\frac{\theta_2}{\theta_1'}$
Δh_c
$\frac{\Delta h_c}{\theta_1'}$
$\frac{\delta_2}{\delta_1'}$
$\frac{\delta_2}{\delta_{os}}$
$r_{3,2}$
$\frac{\delta_3}{\delta_{os}}$
Fig 9
$\theta_4 - \theta_3$
θ_3
θ_4
−
+
+
+
÷
$\frac{\theta_4}{\theta_3}$
Fig 10
$\frac{\delta_4}{\delta_3}$
$\frac{\delta_4}{\delta_{os}} = \frac{\delta_5}{\delta_{os}}$
Fig 12
Eq 19
Fig 8
θ_5
√
$\sqrt{\theta_5}$
$\frac{ST1}{\sqrt{\theta_5}}$
ST1
V_0
−
+
+
ST

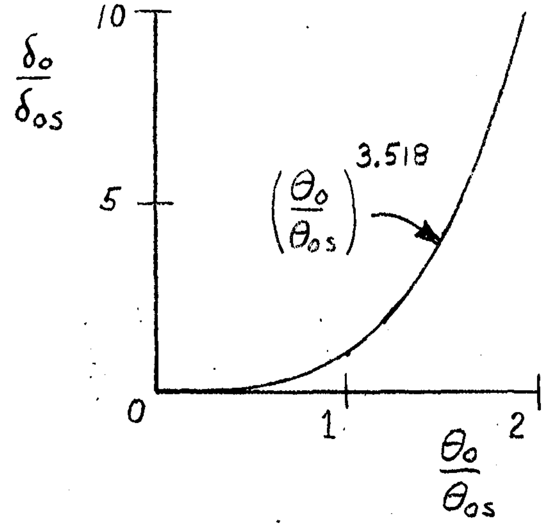
Figure 2.- Dry Turbofan Engine Performance Model.



a) Standard Atmospheric Temperature



b) Inlet Total Temperature



c) Inlet Total Pressure

Figure 3.- Inlet Parameters.

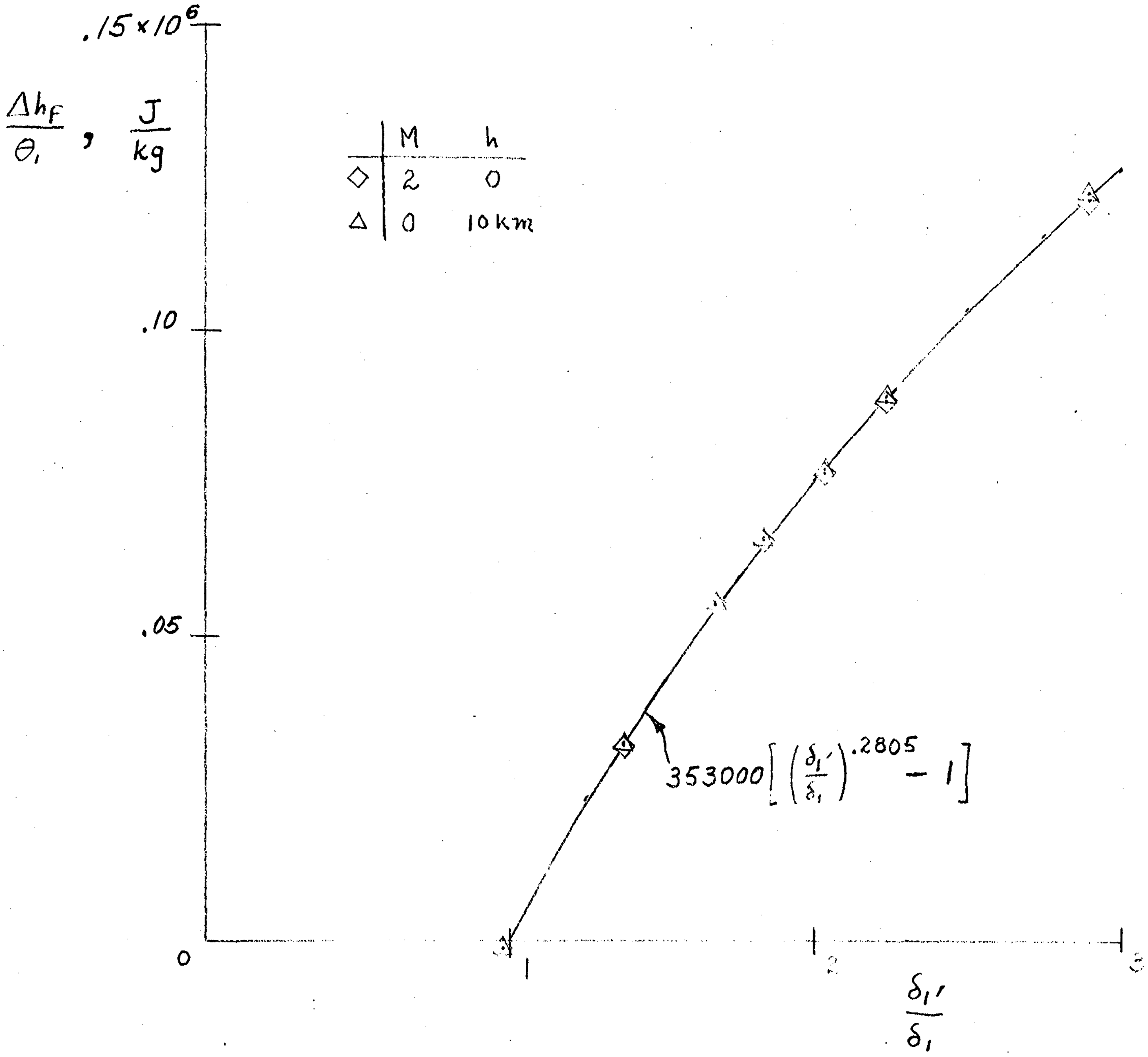


Figure 4.- Fan Enthalpy Change.

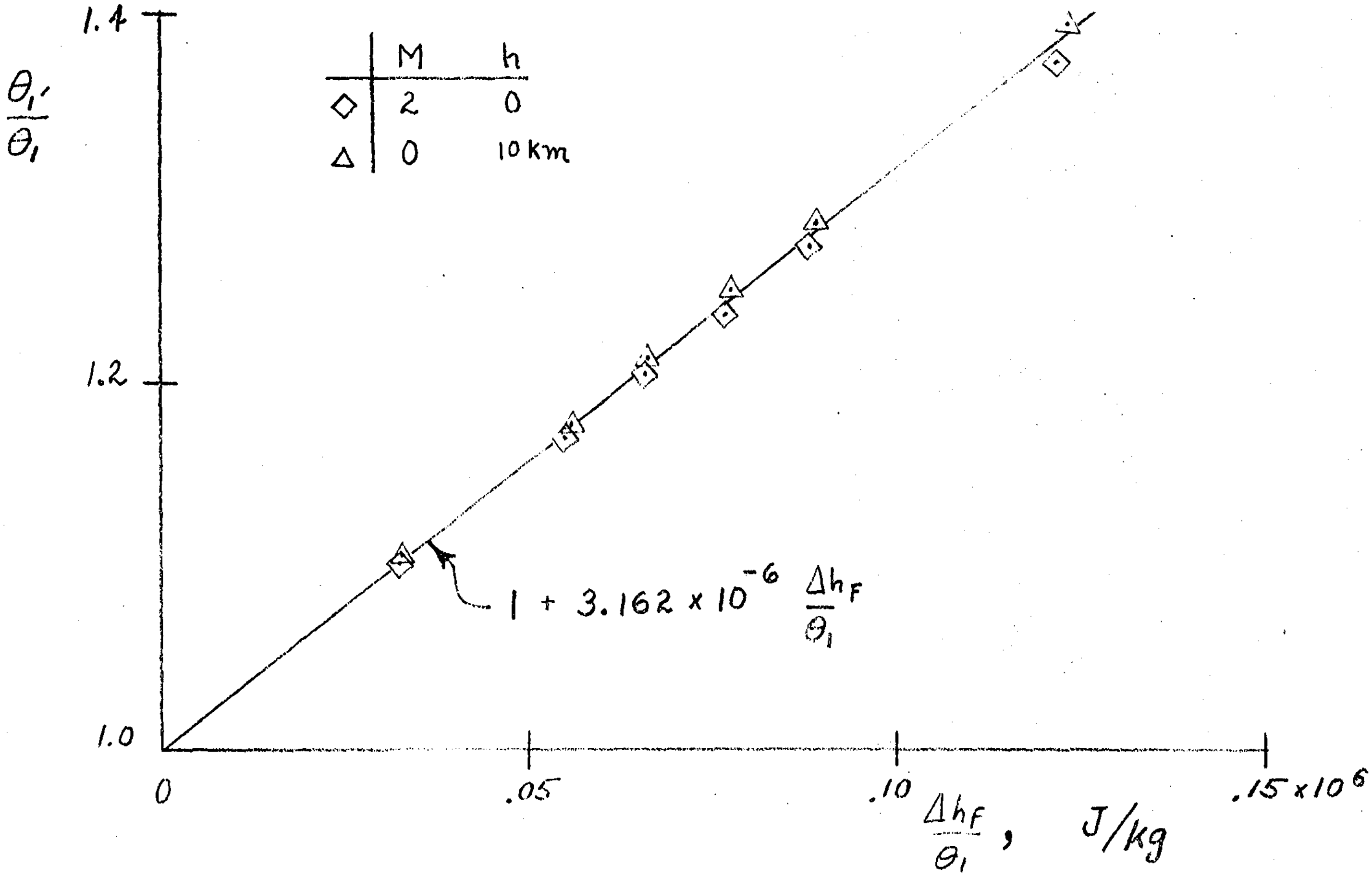


Figure 5.- Fan Temperature Ratio.

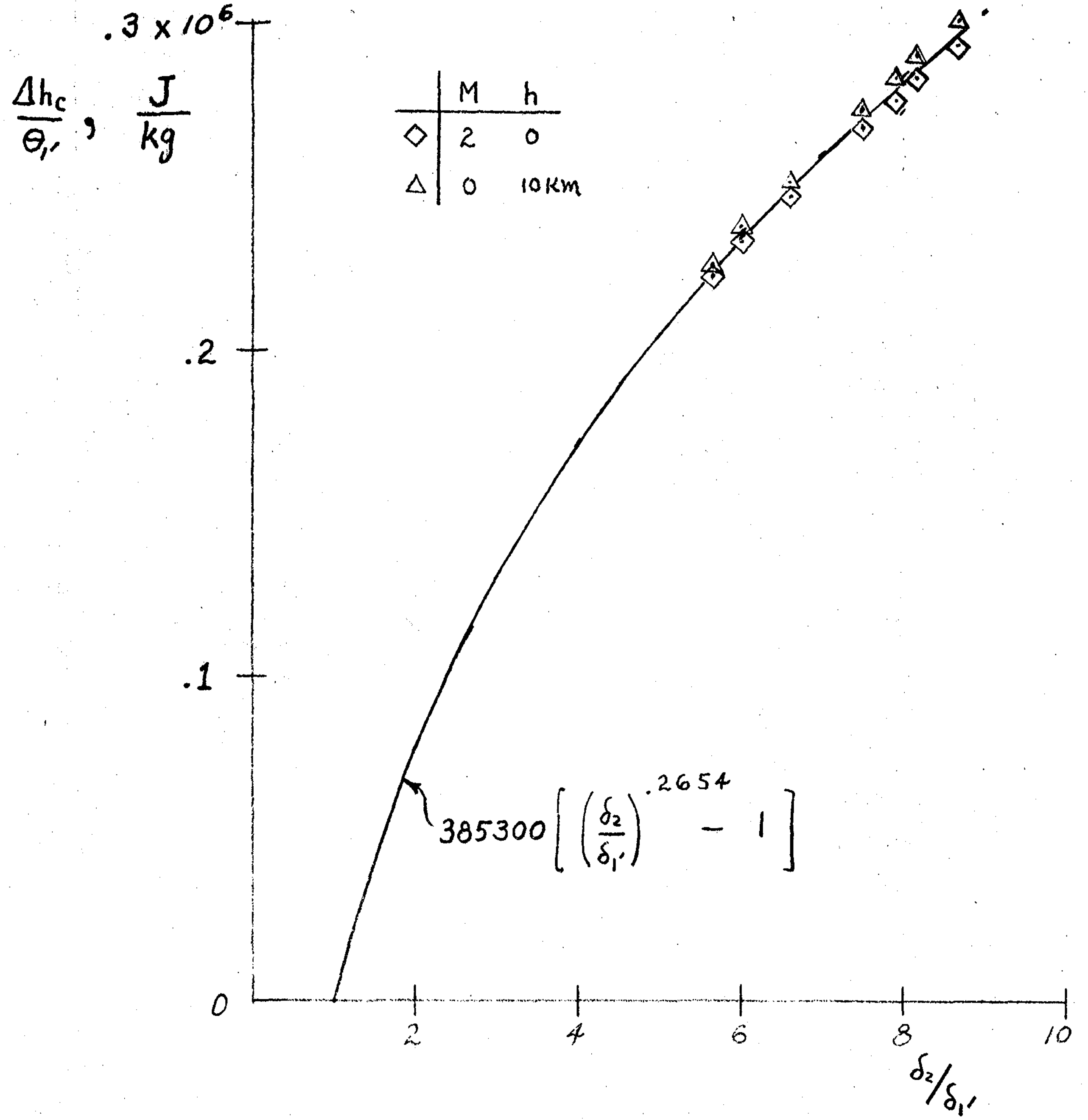


Figure 6.- Compressor Enthalpy Change.

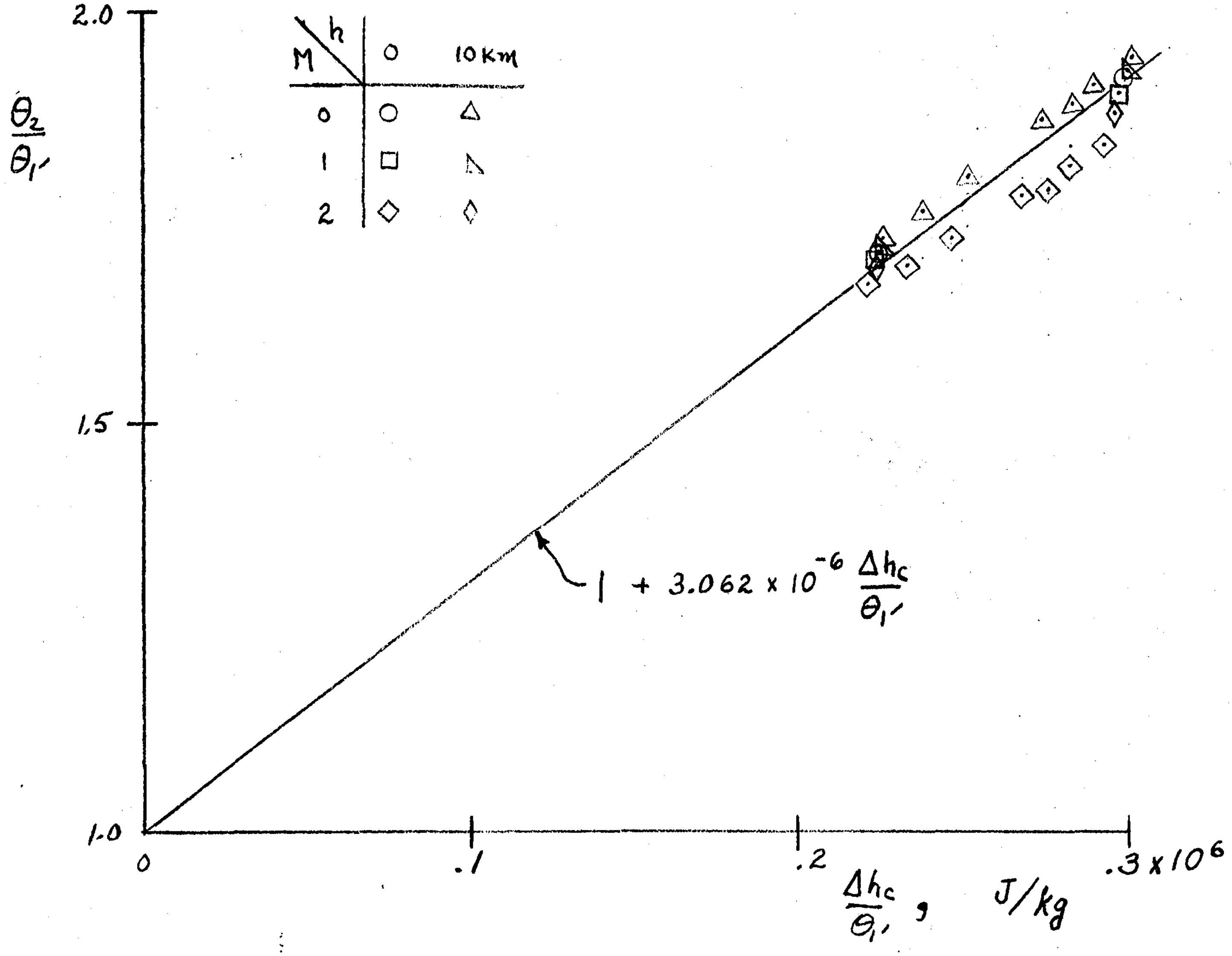


Figure 7.- Compressor Temperature Ratio.

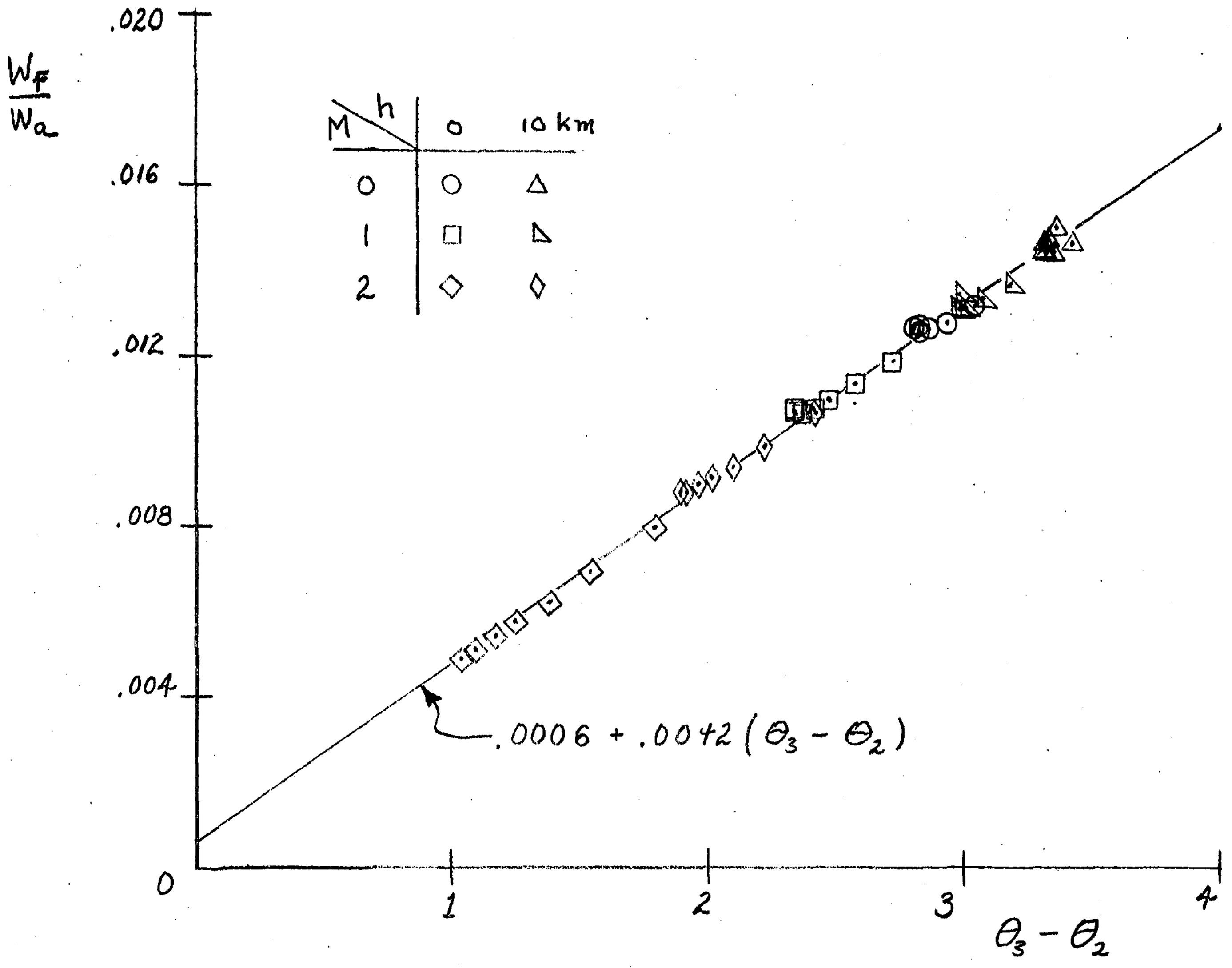


Figure 8.- Fuel-air Ratio.

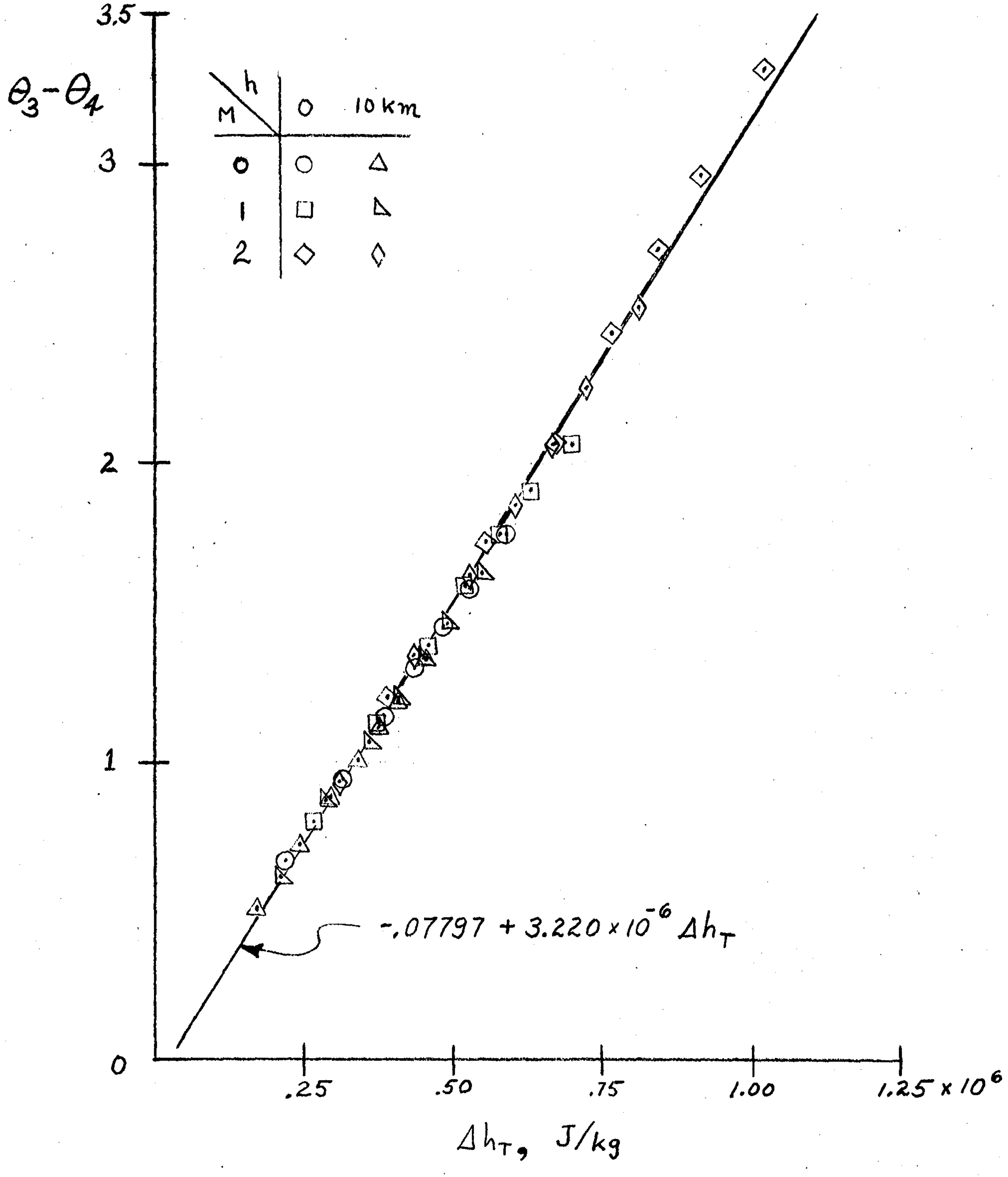


Figure 9.- Turbine Temperature Drop.

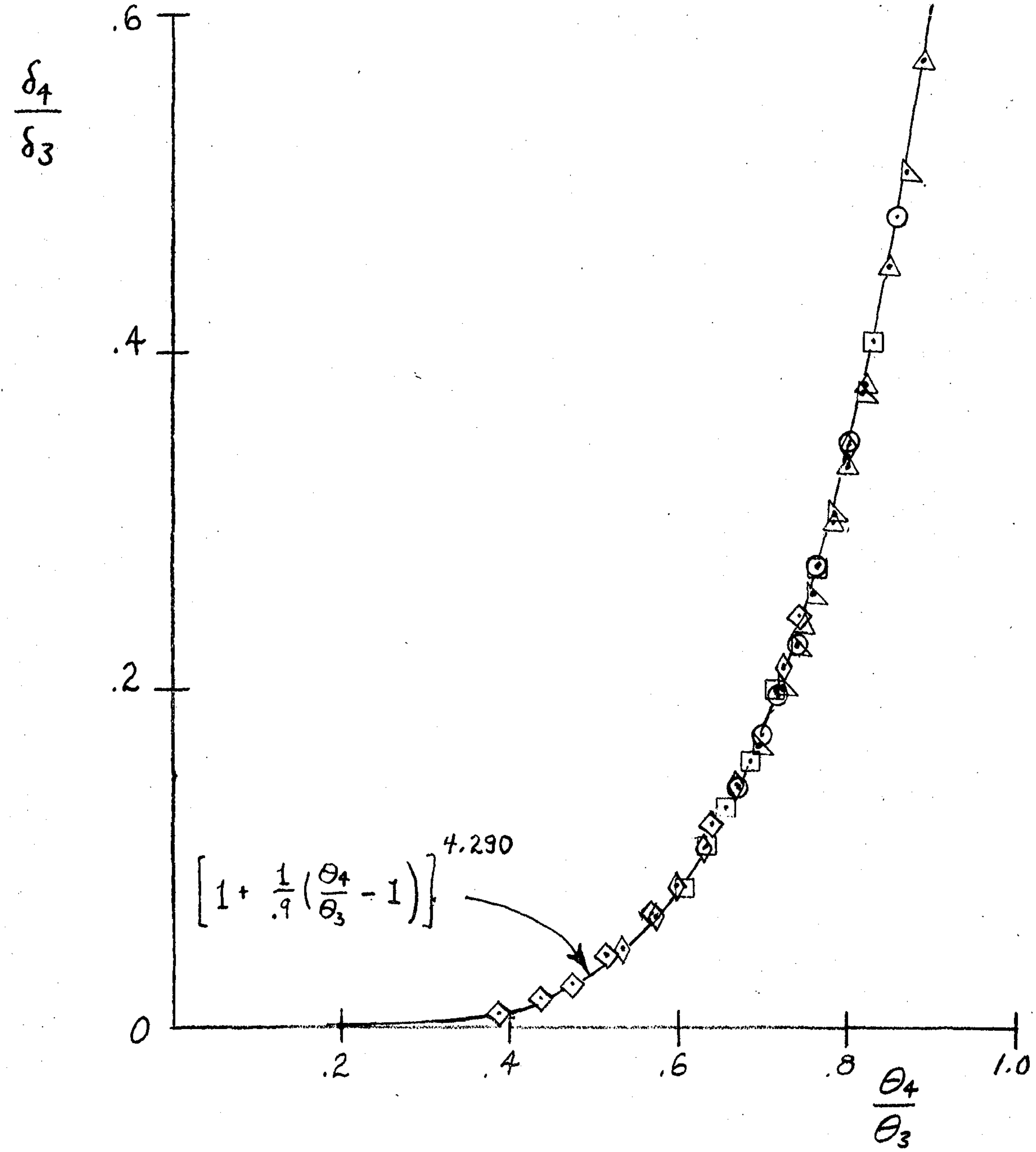


Figure 10.- Turbine Pressure Ratio.

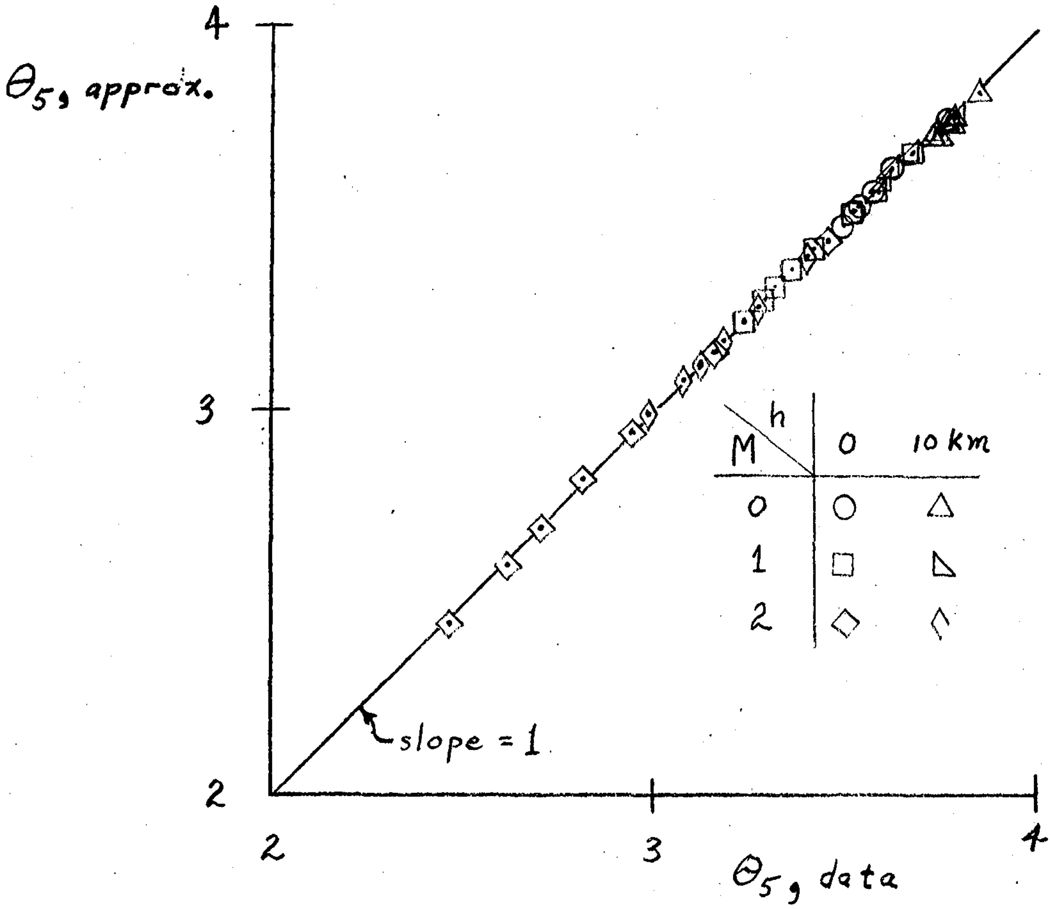


Figure 11.- Turbine Exhaust Temperature.

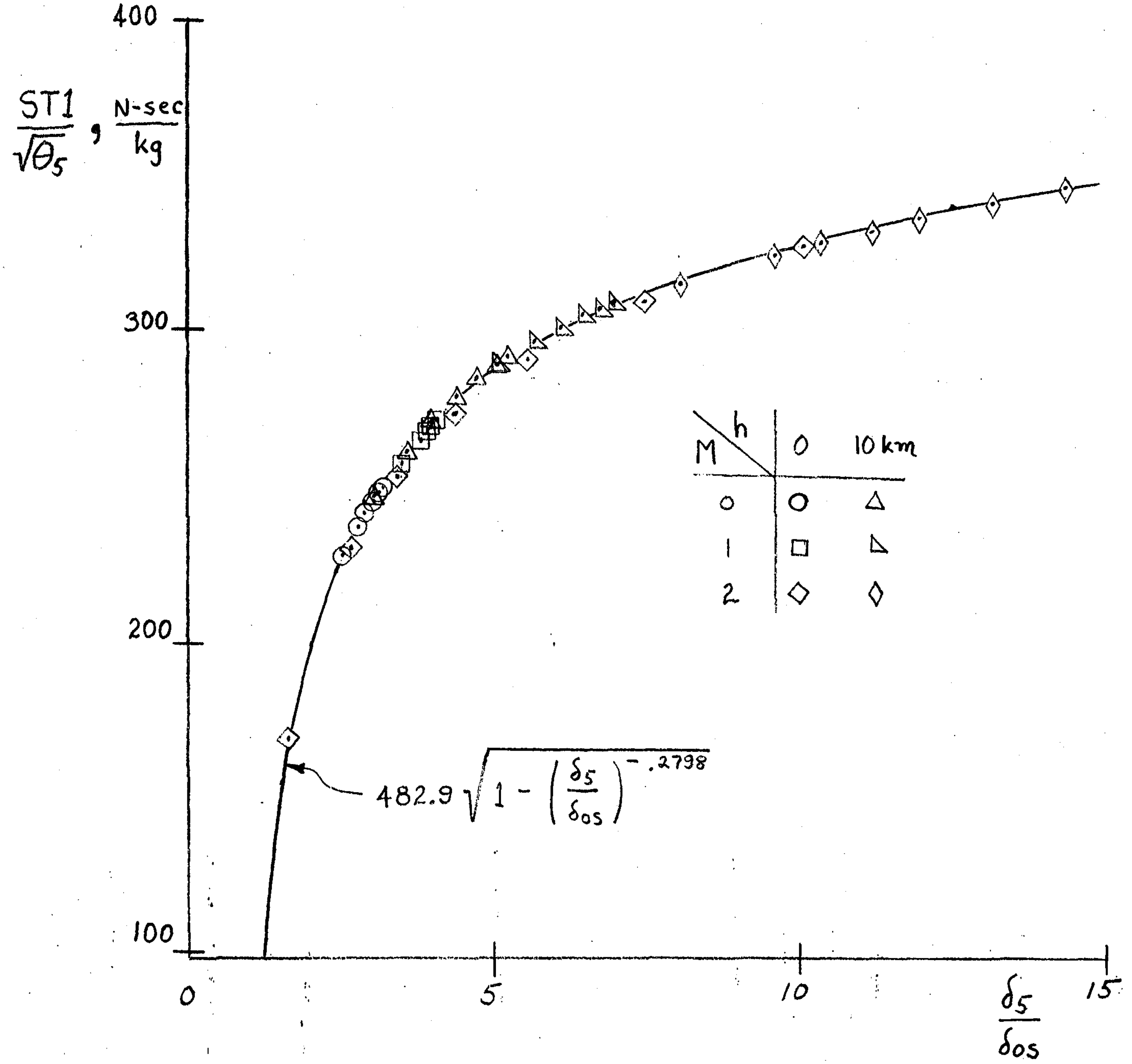


Figure 12.- Main Nozzle.

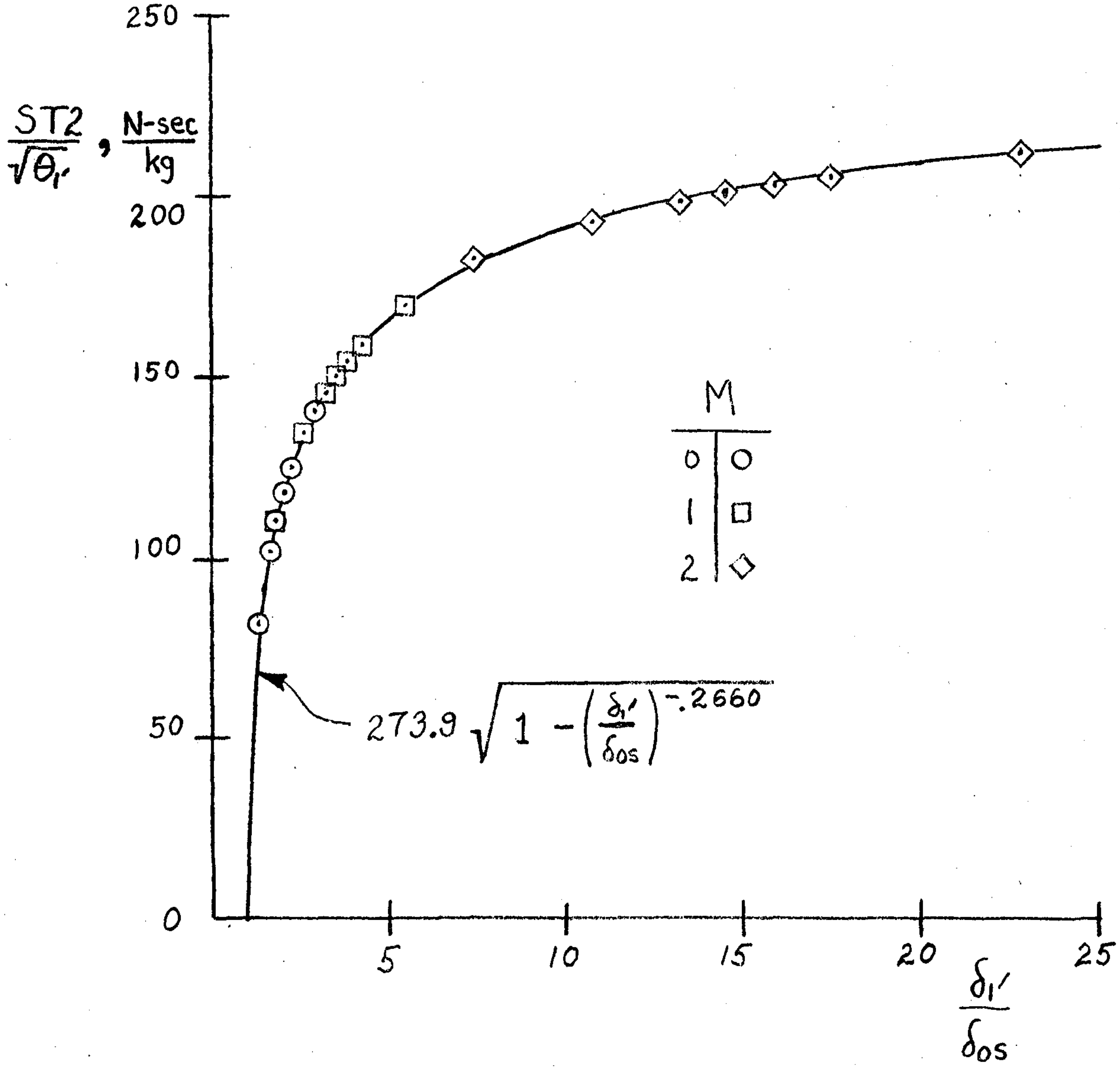
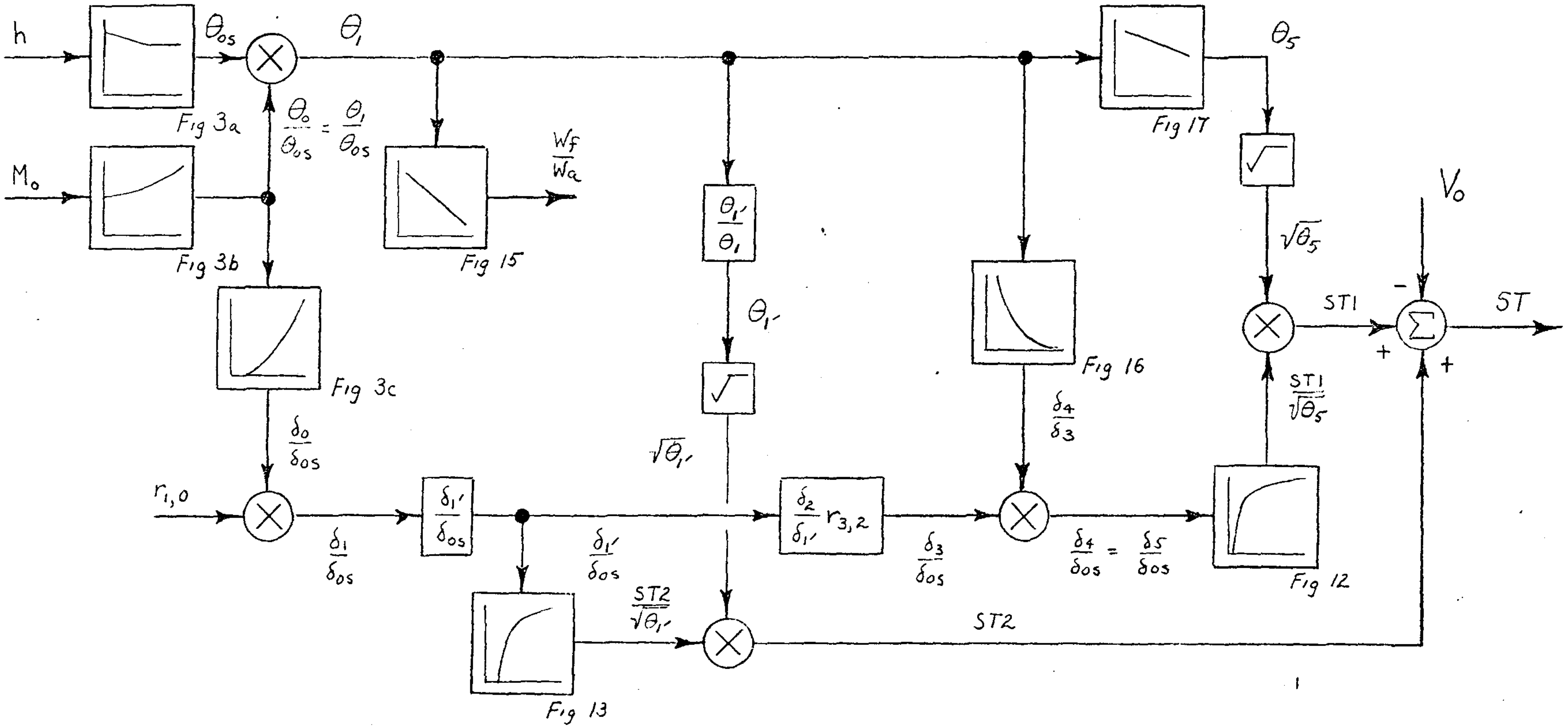


Figure 13.- Duct Nozzle.



h
θ_{os}
θ_1
Fig 3a
$\frac{\theta_o}{\theta_{os}} = \frac{\theta_1}{\theta_{os}}$
M_o
Fig 3b
$\frac{W_f}{W_a}$
Fig 15
$\frac{\theta_{1'}}{\theta_1}$
$\theta_{1'}$
$\sqrt{\ }$
$\sqrt{\theta_{1'}}$
Fig 17
θ_5
$\sqrt{\ }$
$\sqrt{\theta_5}$
V_o
ST1
-
Σ
+
+
ST
Fig 16
$\frac{\delta_4}{\delta_3}$
$\frac{ST1}{\sqrt{\theta_5}}$
Fig 3c
$\frac{\delta_o}{\delta_{os}}$
$r_{1,0}$
$\frac{\delta_1}{\delta_{os}}$
$\frac{\delta_{1'}}{\delta_{os}}$
$\frac{\delta_{1'}}{\delta_{os}}$
$\frac{\delta_2}{\delta_{1'}} r_{3,2}$
$\frac{\delta_3}{\delta_{os}}$
$\frac{\delta_4}{\delta_{os}} = \frac{\delta_5}{\delta_{os}}$
Fig 12
$\frac{ST2}{\sqrt{\theta_{1'}}}$
Fig 13
ST2

Figure 14.- Design-Point Model.

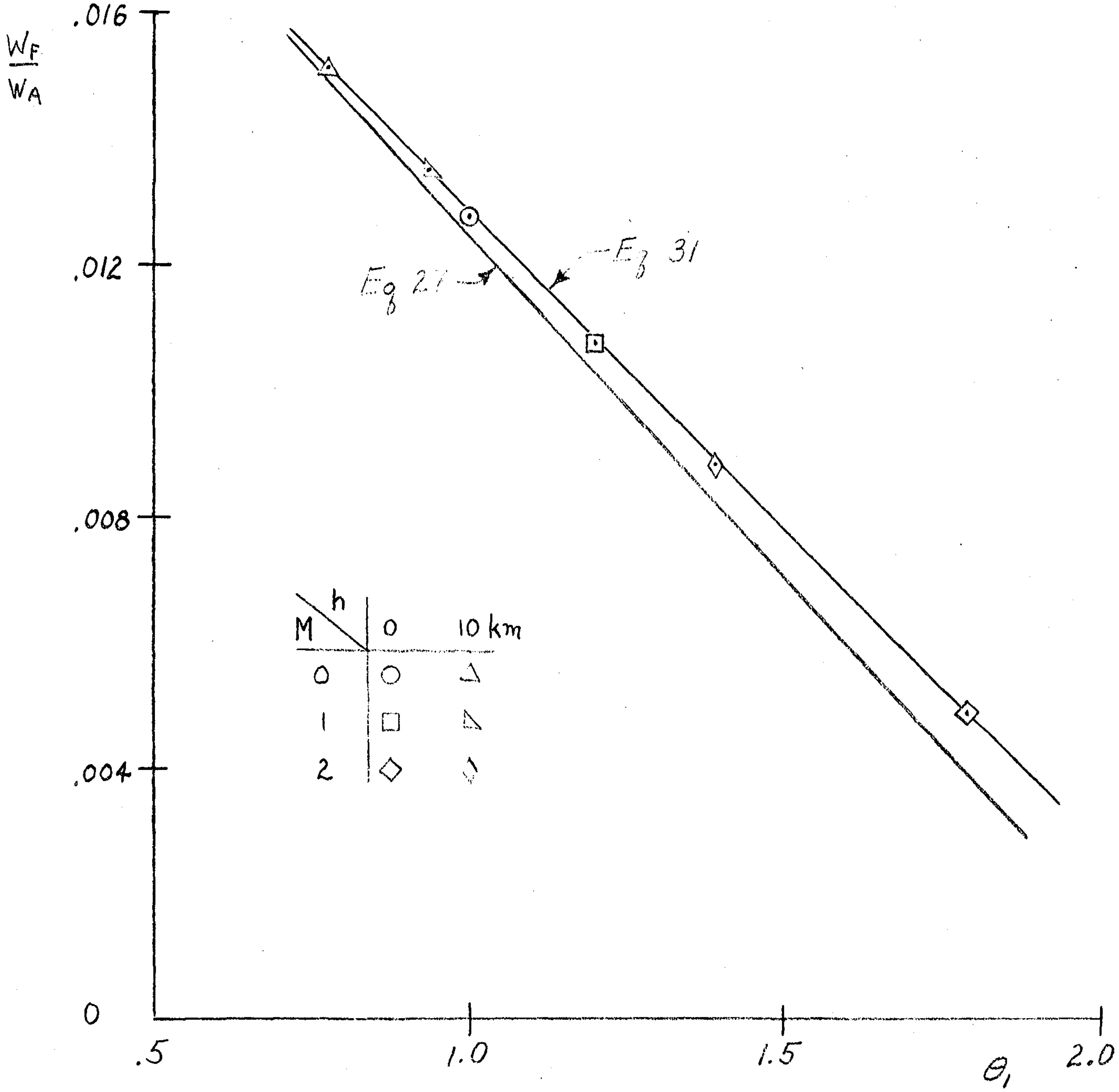


Figure 15.- Fuel-air Ratio.

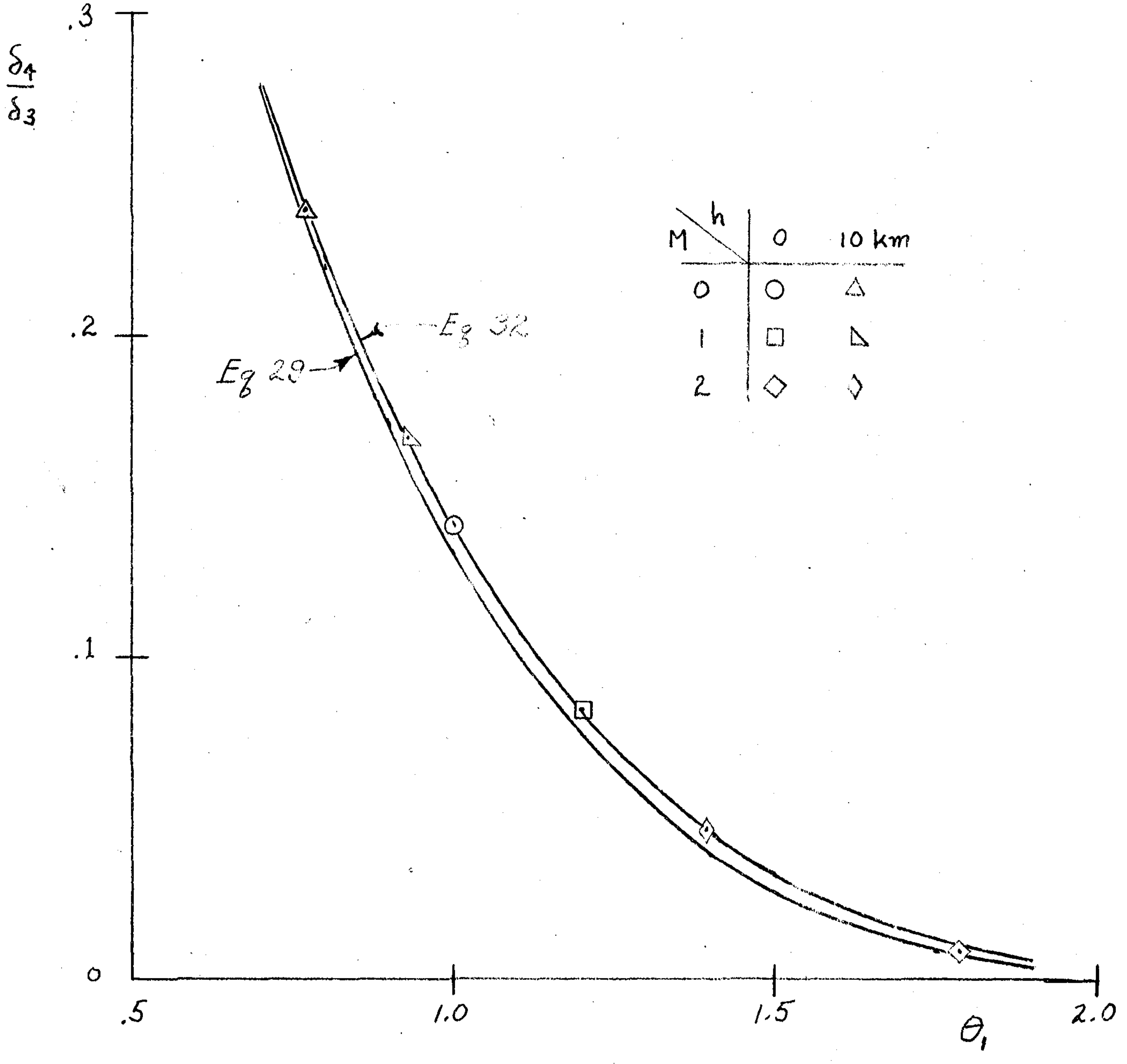


Figure 16.- Turbine Pressure Ratio.

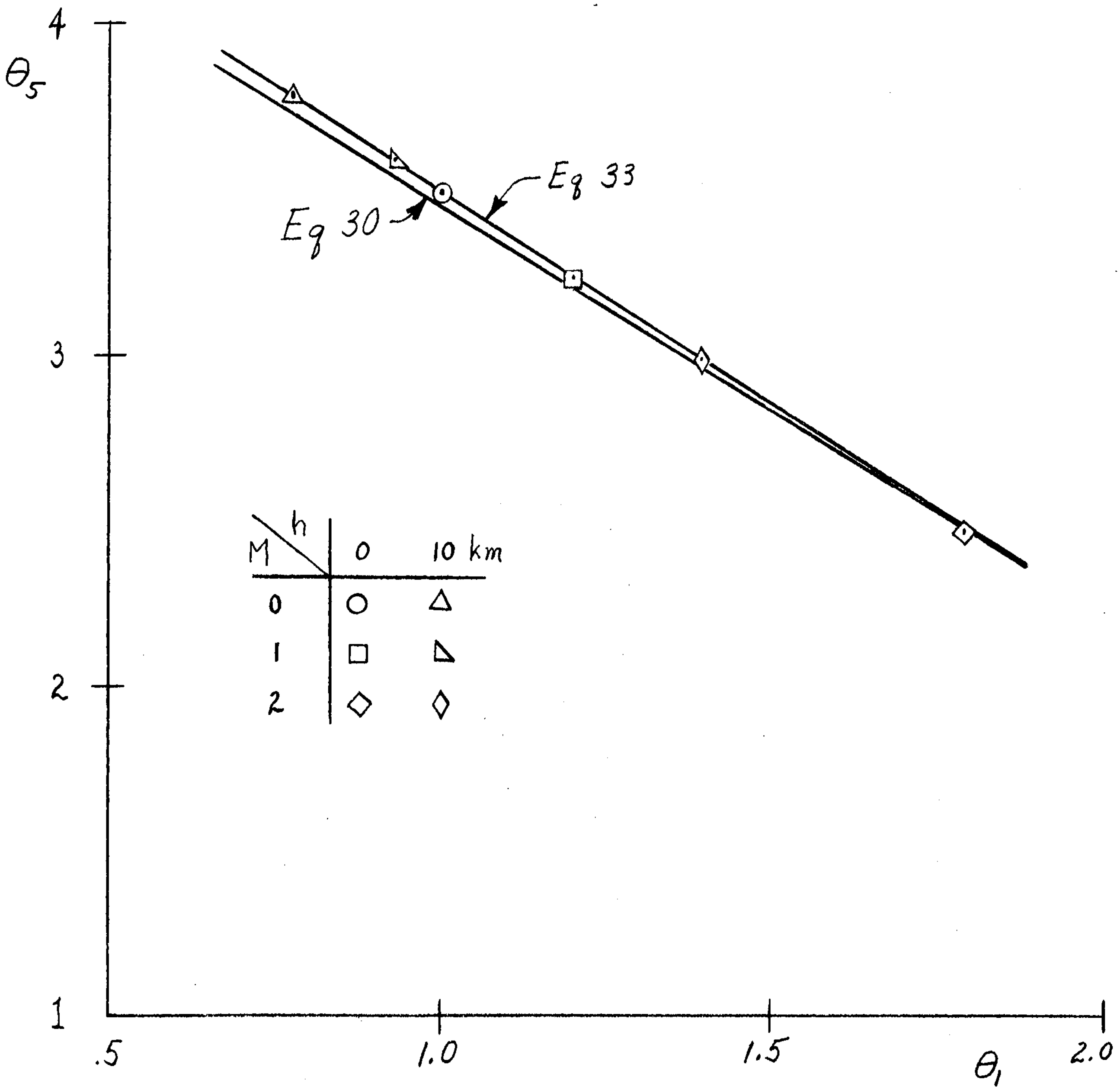


Figure 17.- Turbine Exhaust Temperature.

1. Report No. NASA TM-83204	2. Government Accession No.	3. Recipient's Catalog No.

4. Title and Subtitle Simplified Off-Design Performance Model of a Dry Turbofan Engine Cycle	5. Report Date September 1981 6. Performing Organization Code 505-34-33-02
7. Author(s) Frederick J. Lallman	8. Performing Organization Report No.
9. Performing Organization Name and Address NASA Langley Research Center Hampton, VA 23665	10. Work Unit No. 11. Contract or Grant No.
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546	13. Type of Report and Period Covered Technical Memorandum 14. Sponsoring Agency Code

15. Supplementary Notes

16. Abstract

The specific thrust and fuel-air ratio for a dry turbofan engine cycle were calculated for several power levels over a range of altitudes and Mach numbers. The engine has a design fan pressure ratio of 2.9, compressor pressure ratio of 8.0, and bypass ratio of 0.6. Nominal engine component curves were picked to approximate the calculated data to construct a simplified model of the off-design performance of the engine. The model was then used to construct a simplified design-point engine model for the full-power condition.

17. Key Words (Suggested by Author(s)) engine model turbofan steady state performance	18. Distribution Statement Unclassified - Unlimited Subject Category 07

19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 32	22. Price* A03

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